Scramjet Thermal Management
(Tenue thermique des superstatoréacteurs)

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ABSTRACT

The lecture of the RTO-AVT-VKI course "HIGH SPEED PROPULSION: ENGINE DESIGN – INTEGRATION AND THERMAL MANAGEMENT" was given at the Von Karman Institute in September 2010 and in the Wright State University in December 2010. The present lecture was focused first on the estimation of the heat loads on a scramjet or a dual-mode ramjet. Solutions to sustain such high energy are secondly described and the lecture addresses how to combine materials, cooling techniques and system requirements. A simple exercise was proposed to the attendees as an illustrative understanding of regeneratively cooled generic scramjet flying at Mach 7. The results are based on published information, and additional details, presented to the lecturers, can be found in the references.

RESUME

Le stage RTO-AVT-VKI "HIGH SPEED PROPULSION: ENGINE DESIGN – INTEGRATION AND THERMAL MANAGEMENT" fut donné à l'Institut Von Karman en septembre 2010 et à l'Université Wright de Dayton (Ohio, USA) en décembre 2010. Le présent cours aborde la tenue thermique. Il est basé sur des informations publiées, et des détails supplémentaires, présentés aux personnes qui suivaient le cours, pourront être trouvés dans les références bibliographiques. Trois aspects ont été discutés, avec éventuellement quelques calculs simples de compréhension : quelles sont les charges thermiques à supporter ? comment faire pour y résister ? quelles technologies font l'objet de travaux dans le monde ?

CONTEXT

Scramjets or Dual Mode Ramjets for High Speed Atmospheric Flight

NASA, DOD, the U.S. industry and global community have studied scramjet-powered hypersonic vehicles for over 40 years [1] [10] [22].

In a large part of the flight regime, the air-breathing mode appears to be a good possible solution for future Reusable Space Launchers (RSL). Dual-mode ramjets have been studied to propel such TSTO (Two Stage To Orbit) or Single Stage To Orbit (SSTO) vehicles. For example, in the scope of the French PREPHA program, the study of a generic SSTO vehicle led to conclusion that the best type of airbreathing engine could be the dual-mode ramjet (subsonic then supersonic combustion).

Airbreathing launchers could typically use hydrogen-fuelled DMR. Less energetic fuels like hydrocarbons could also be used at a Mach number lower than 8, to take advantage of their higher density. In this case, the engine must be able to manage two different fuels.

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15. SUBJECT TERMS

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High speed aircraft could use hydrogen, liquid hydrocarbon (kerosene or endothermic fuels like JP7) or methane.

Hypersonic military applications and associated DMR or scramjets are typically associated with liquid hydrocarbons and a maximum Mach number of 8 (missiles).

For most of these applications, the scramjet is not used alone but is generally one particular operation of a dual-mode ramjet (DMR): a ramjet able to operate in subsonic combustion (for a flight Mach number between 1.5 and 6 for example) then in supersonic combustion (for a flight Mach number over 6 for example).

In a supersonic combustion ramjet, while heat release is equivalent to cross section decrease, the Mach number decreases (but remains supersonic) while the pressure and the temperature increase. 1D and 3D computational tools are used in connection of testing to optimize the engine. An example is given in [13], for a supersonic combustion chamber tested under Mach 6 flight conditions for two fuelling conditions.

Two main challenges are attached to scramjet or DMR:

1) To ensure a sufficient aeropropulsive balance (which could lead to particular flow-path contours and to have variable geometries).
2) To ensure sufficient thermal and mechanical strength (topic of the present lecture).

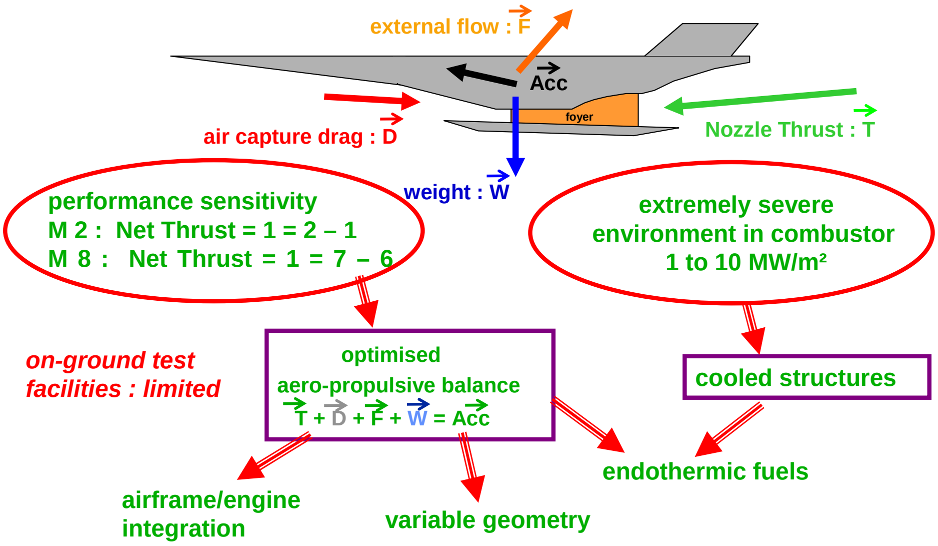
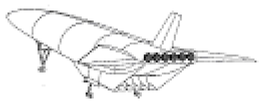


Figure 1: The two main challenges for scramjet design.

Two main ways of approach are possible for the DMR: a fixed or a highly variable geometry. The propulsive performance (thrust, consumption) of the DMR have to be optimised, computed and at-best demonstrated. But a major concern is the capacity to build such an engine, and to estimate and demonstrate its robustness and its weight.

The scale factor is also a big issue, from small vehicles like missiles or X51 demonstrator up to huge scramjet for high speed aircraft or future reusable launch vehicle. The capture area Ac determines the air ingested then fuel mass flow (that can be used to actively cool the engine) while the wetted area Aw is the one to be cooled : we should maximise the first one and minimize the second one.

Table 1: Size and characteristic areas of scramjet propelled vehicles.

	Lenght	Initial weight	Captation area	Engine wetted area
Missile or small experimental vehicle 	6 m	1,5 tons	0,15 m^2	1,5 m^2
Reusable space launcher SSTO 	70 m	500 tons	30 m^2	100 m^2

An intermediate size has sometimes been studied (30 meters 30 tons vehicle).

The variable geometry can be limited to a single degree of freedom, of rotating or –the example chosen here- in translating a cowl that modifies simultaneously the minimum cross section of the air intake and the combustor geometry [8].

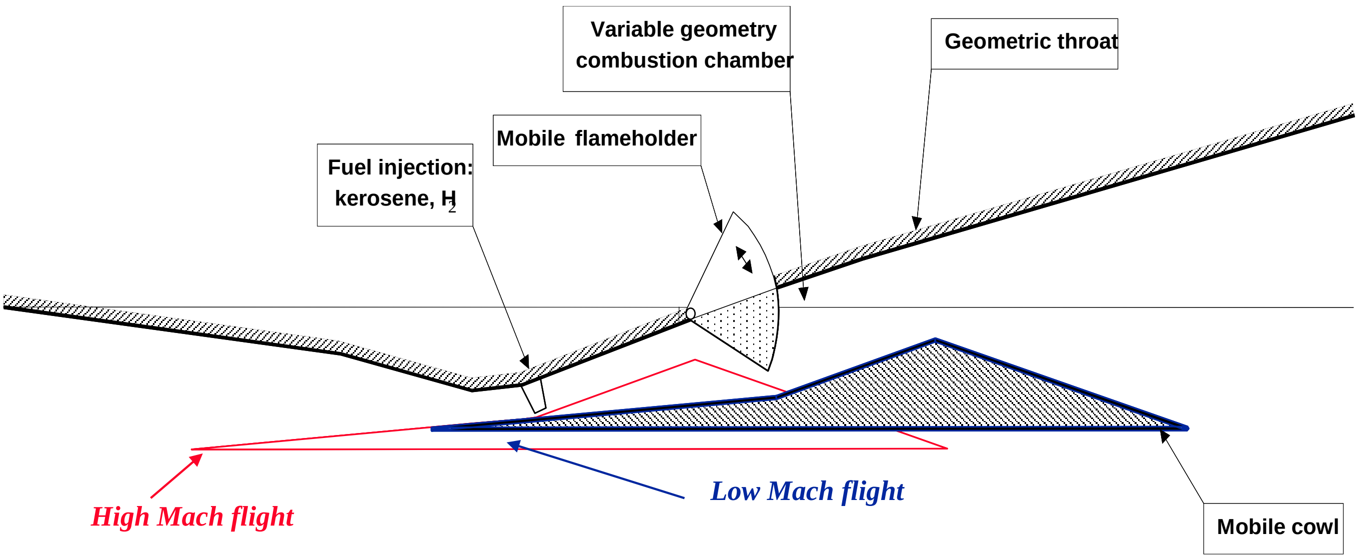


Figure 2: Translating cowl DMR concept.

The computed flow-field in such an engine served as a reference in the present lecture, to understand the thermodynamic parameters in the combustor or to estimate ‘by hand calculations’ the local heat transfer.

The figures below give these reference values for Mach number, static pressure and static temperature [69].

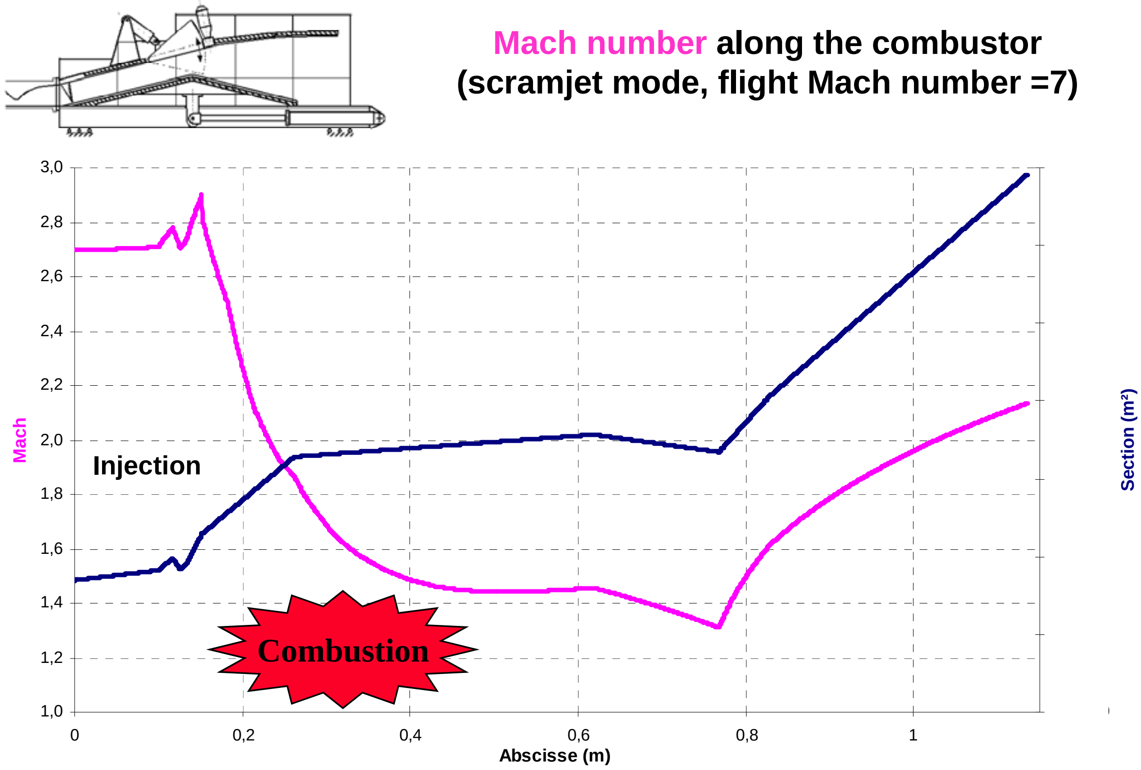


Figure 3: Local Mach number in a scramjet combustor [69].

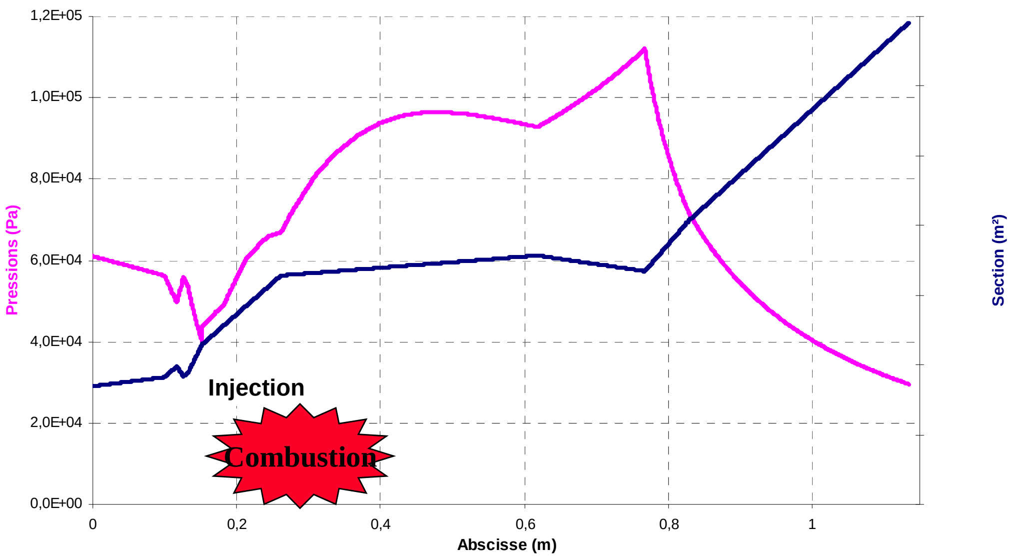


Figure 4: Pressure contour along the scramjet [69].

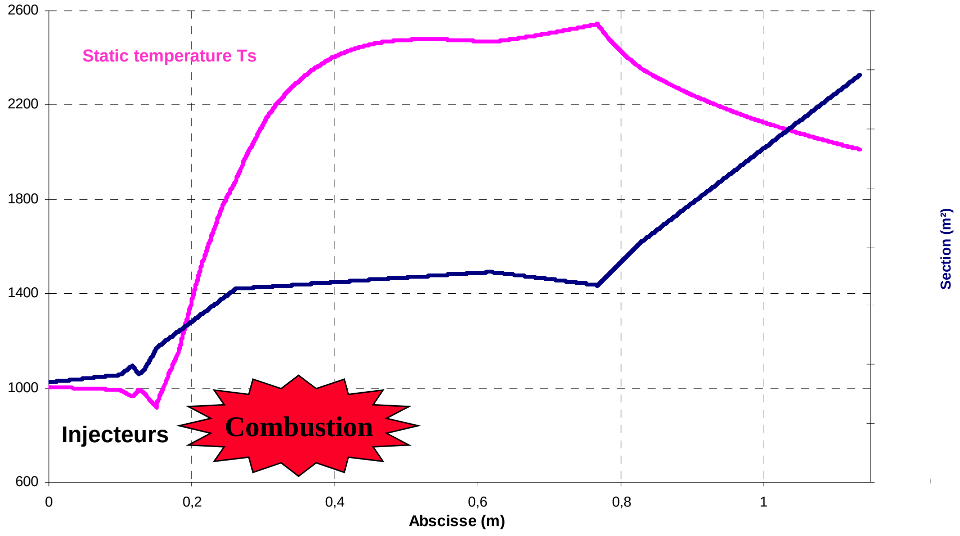


Figure 5: Static temperature contour in the scramjet [69].

Flying Above the "Thermal Barrier"

If Mach 1 is the "sound barrier", Mach 5 is generally considered as the "heat barrier", because of the thermodynamic effect on the air and mostly because of the fusion of stagnation points in classical materials due to the high kinetic heating [37].

Stagnation temperature increases with the square of the flight Mach number.

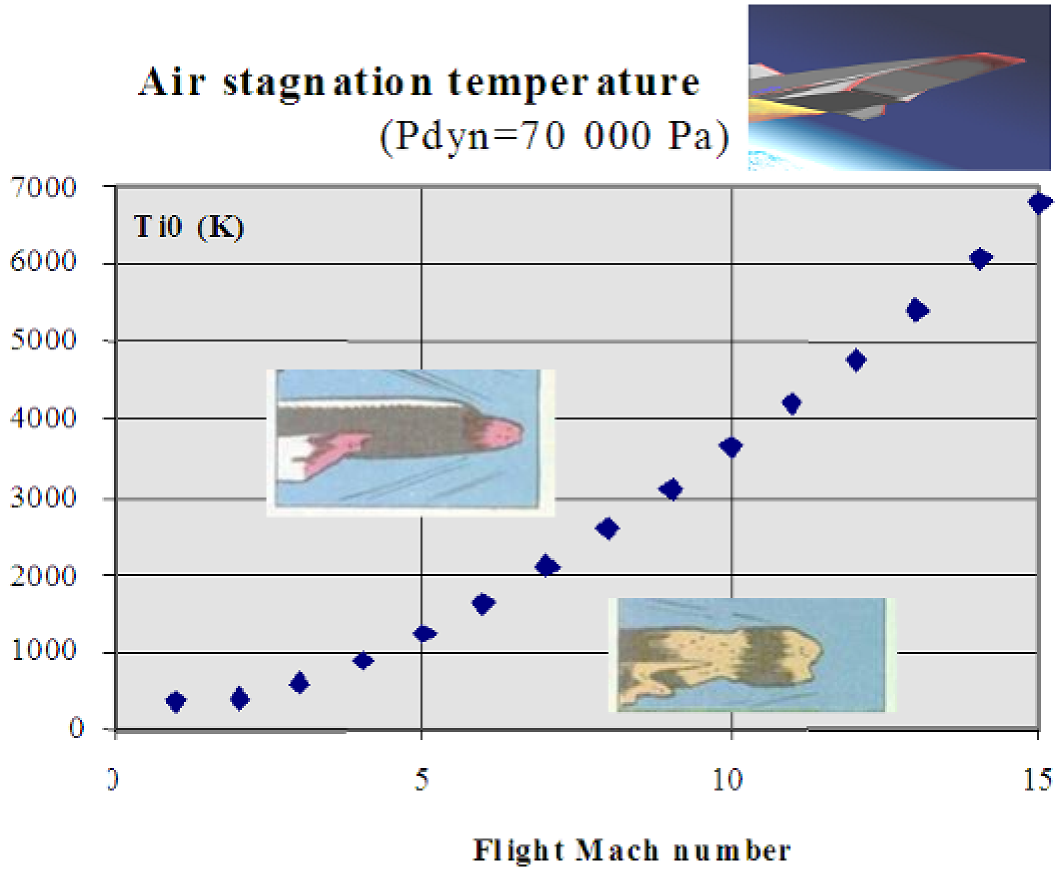


Figure 6: Dramatic increase of stagnation temperature with flight Mach number.

The vehicle becomes more and more hot, but can radiate to the cold atmosphere and limit in temperature, et least far from the stagnation points. The engine, in opposite, is a closed box and can not evacuate so simply the heat, and moreover the combustion adds energy.

To solve this issue, the engineering team has to estimate the heat loads and then to design solutions to afford them.

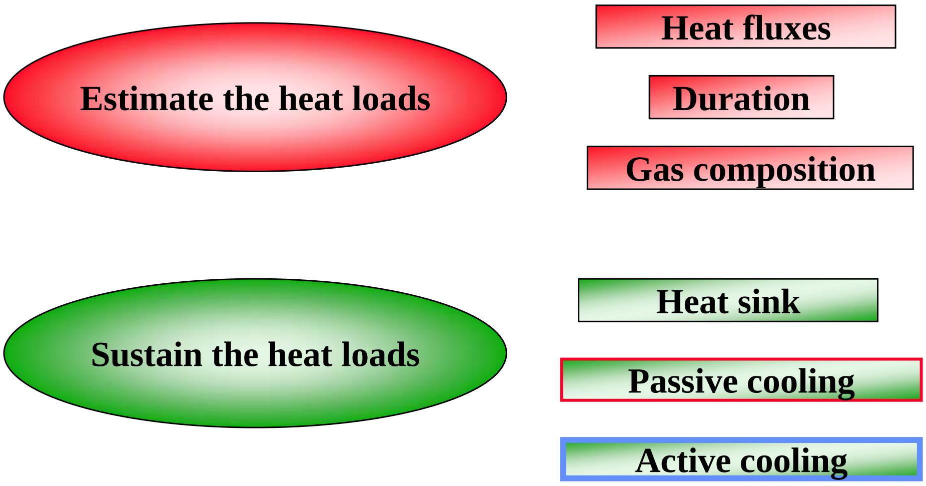


Figure 7: Scramjet thermal management.

ESTIMATE THE HEAT LOADS

Generalities

The heat loads have to be estimated and checked, and associate with time (duration) and also gas nature (O2 ? H20 ? CO2 ?) in contact with the hot wall of the scramjet.

The heat loads can be measured with different techniques, but the analysis and the preparation of the acquired data has to be done carefully. Inverse techniques and measurement analysis require special awareness; see for example [19].

The heat loads (hot gases recovery temperature Taw, convective heat transfer coefficient hg for each trajectory point) are mainly investigated with several semi-empirical methods.

Heat transfer rate density (W/m²)

$$q_{cv} = h \times (Tath - Tp)$$

$$Tath \approx Ti$$

r : recovery factor (typical value : r= 0,9 for turbulent boundary layer)

$$hath = hs + r \times \frac{V^2}{2}$$

Stanton number :

Reynolds analogy : $s = \frac{St}{Cf/2}$

$$St = \frac{q_{cv}}{\rho_e \times V_e \times (h_{ath} - h_p)}$$

Figure 8: Heat flux density (convection).

Heat Loads Estimate for walls – Industrial Methods

The CFD can more and more be used to determine the heat loads and especially the local heterogeneities. An example of 3D computational results of a high speed scramjet (CEDRE code with Reynolds-Averaged Navier-Stokes equations solver, regular turbulence model and medium-detailed kinetic modelling of fuel/air combustion) shows how the heat loads can vary with 3 increasing values of the injected equivalence ratio [20].

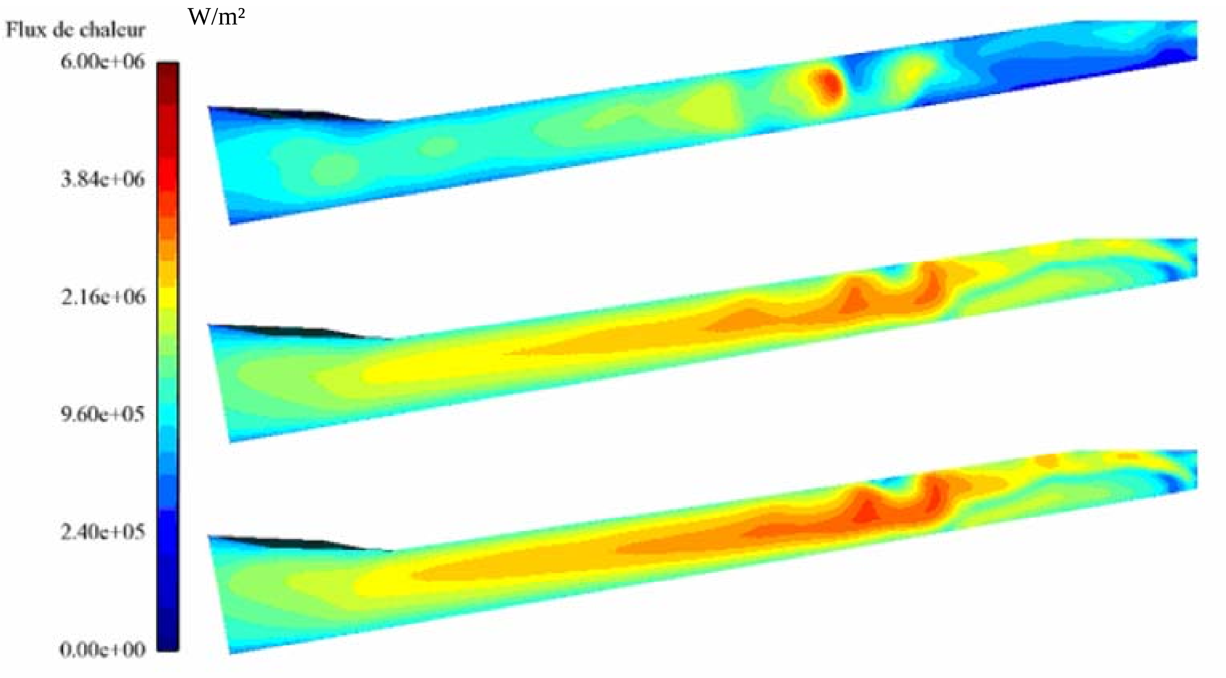


Figure 9: 3D reactive computation of scramjet: heat flux density [20].

As an other example, the 1-D code PUMA (French acronym for One-dimensional Program for Analysis of Aerothermochemistry) is used extensively by MBDA for advanced studies of scramjets and to provide a first analysis of experimental results such as *CHAMOIS* one [13]. The PUMA code is also used to estimate the heat fluxes along the DMR combustor. The heat transfer coefficient is estimated using semi-empirical Colburn law in a duct. To take into account compressibility effects and non-adiabatic wall conditions, the Spalding and Chi equation has been used [21].

For several CHAMOIS scramjet test results, the PUMA-computed heat fluxes have been compared with the post-processed results of the heat-flux-meters. Because CHAMOIS is a heat-sink mock-up, the wall temperature varies with time and varies spatially along the duct. Inverse methods are used to take into account the delay due to heat propagation between the two thermocouples of the heat-flux-meter. The results obtained by the above formula with PUMA computation were in good agreement with experimental data.

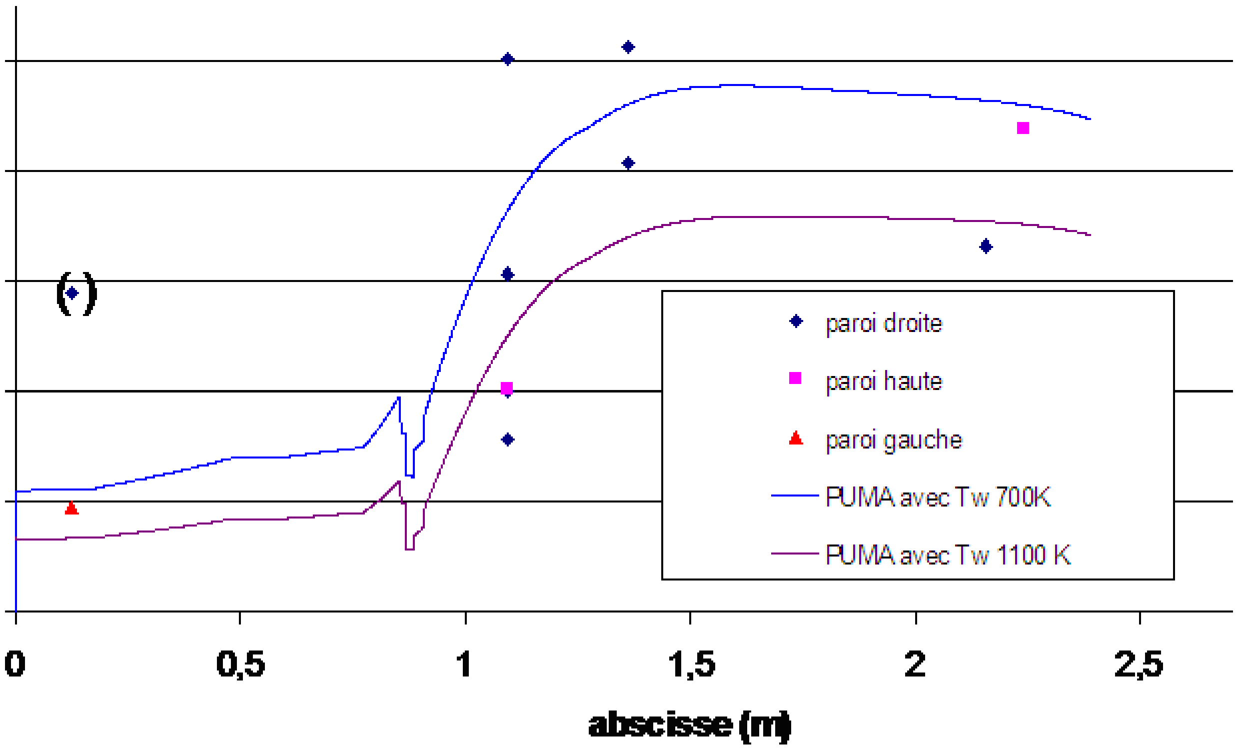


Figure 10: Computed and measurement heat flux for CHAMOIS scramjet test [21].

The radiation of hot species is generally neglected in scramjet environment, considered as "hidden" in the uncertainty coefficient (of 30 % for example) we will add to the convective heat flux estimating. Nevertheless, the overall level of gas radiation (mainly H2O, CO2 and CO) can be estimated and compared with the convective heat fluxes. For that, the modified shack formulae can be used [36].

Heat Loads Estimate for Walls – Rapid Estimate

For rapid estimate, it is possible to use the Colburn formula. The heat transfer coefficient is then taken from the Nusselt definition. Turbulent flow is liable to be assumed in scramjets and dual-mode ramjets.

Heat Loads Estimate for Struts Leading Edges

The heat loads (hot gases recovery temperature Taw, convective heat transfer coefficient *hg* for each trajectory point) are mainly investigated with several semi-empirical methods.

For a typical SSTO trajectory at a constant dynamic pressure of 60000 PA, and for typical foreboby and air intake assumptions [6][7][8], the stagnation heat flux density $hg \times (Taw - Tw)$ on the leading edge of radius *re* can be estimated from the following formulas up to a flight Mach number M_f from 5 to 15:

$$hg = \frac{(54 \times Mf \ - 76)}{\sqrt{re_{\,(\mathrm{m})}}} \; (\mathrm{W/m^2/K})$$

$$\mathrm{Taw}_{\,(\mathrm{K})} = 0{,}8196 \times \mathrm{M_f}^3 + 42{,}705 \times \mathrm{M_f}^2 - 4{,}5729 \times \mathrm{M_f} + 318$$

In case of shock/shock interaction, the local heat transfer could sometimes be lower and often much higher (type III or IV interactions). An example is given in the Schlieren below, for a leading edge radius of 3mm and a free stream Mach number of 4.96 [17].

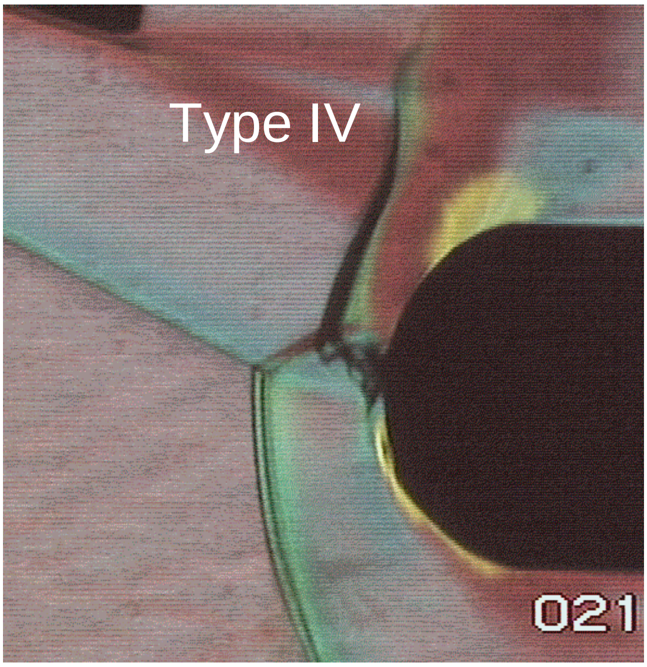


Figure 11: Type IV interaction studied at CNRS SH2 wind tunnel.

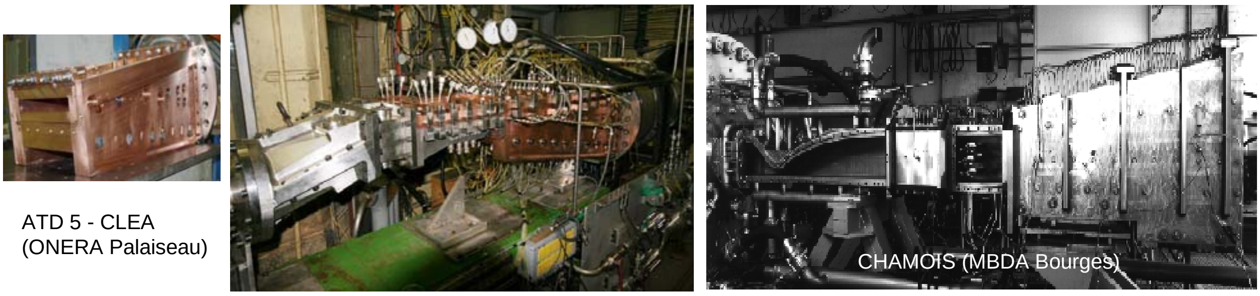
Analytical formula and specific computational tools have been specifically set and are used to estimate the overloading on 1.5 to 3 mm radius leading edges [17].

SUSTAIN THE HEAT LOADS

Different techniques can be employed, that were generally already used on other systems [23][25][27] [38][39] [45][58].

Heat Sink Structure

The heat sink solution is very often preferred for combustion process experimental studies, where a test duration of less than 15 s in enough to document different combustion results while varying operating conditions (equivalence ratio change generally). The walls are either on stainless steel (NS30 – A310 for example) like CHAMOIS or copper-type material like the CLEA engine in ATD5 at ONERA [20]. A Thermal Barrier Coating can be added on the two solutions.



Copper-type with or without thermal barrier

Stainless steel with or without thermal barrier

Figure 12: Examples of heat sink scramjet combustion mock-up for ground testing.

For experimental vehicles, if the test duration is small, the same technique is used (X43-A [22] [68] or LEA [41]), while X51 uses a regeneratively cooled engine, for several minutes of operation [52].

Ablative thermal protection or insulation are used for ramjets [45] but the temperature and heat fluxes levels in a scramjet require other solutions.

Radiation Cooling

$$q_{Ray} = \varepsilon \times \sigma \times (T_p^{\ 4} - T_s^{\ 4})$$

$$\sigma = 5{,}67.10^{-8} W.m^{-2}.K^{-4}$$

If the structure can sustain very high temperature with a good conductivity and emissivity, at moderate heat loads, it could be possible to use the outside radiation, as it was made in the 90's at Moscow Aviation Institute under Mach 6 flight conditions. The combustor was made of Niobium.

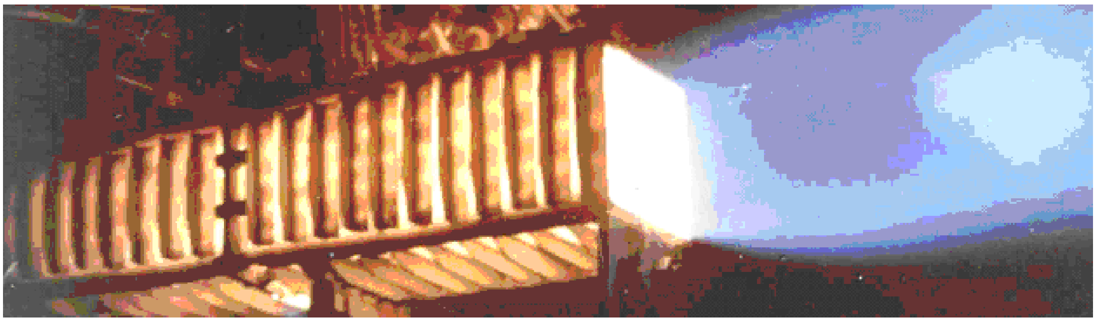


Figure 13: Massive Niobium scramjet combustor under Mach 6 flight conditions testing (MAI).

The stiffeners have also an *ailette* (fin) thermal effect.

Active Cooling 1: Regenerative Cooling – Fuel Cooled Structures

Here the fuel is used to cool the engine, before to be injected in it. The coolant is generally directly the fuel, secondary loops systems (indirect cooling through an additional coolant in closed loop) are not used for scramjets because of complexity and weight.

Several circuits of active cooling systems had been compared to ensure good behaviour of the engine walls, with respect of the combustion-required fuel mass flow in many “paper” studies. Many configurations of cooling have been envisaged, such as series of channels of rectangular shape or pin fins channels. The pin-fin circuit (Figure 14) was confirmed in different studies as more efficient [26] than the more classical machined-channels.

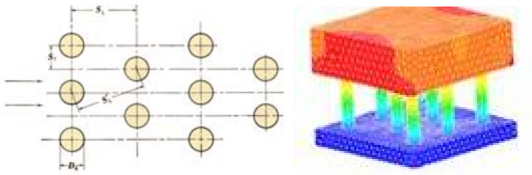


Figure 14: Pin fin configuration of cooling channel.

One of the interest of the pin-fin circuit is the easy management along the walls of the DMR combustor, which is often diverging. Nevertheless, this promising technique of pin fin cooling seems often more difficult to be used for manufacturing reasons.

The interest of using composite structures (able to operate at temperature over 1800 K in oxidizing environment and with a typical density of 2) have been demonstrated thanks to several analytical and computational studies and to actual technology experimental testing (benefit in weight, benefit in thermal capability, benefit in injection strut drag, …)[50]. It will be addressed later in the present document.

Parametric studies have been performed, with the trajectory effect: the increase of the flight dynamic pressure has been demonstrated as a benefit for such a hypersonic airbreathing vehicle (it is the opposite in case of heat sink or radiative cooling).

Active Cooling 2: Injection, Effusion, Film

It is also possible to use an injection cooling technique through a slot (film cooling) or a multi-perforated or porous wall (effusion, transpiration, …). Coolant can be air (if it is still enough cold to cool the wall to stay under its material limitation) or fuel (a small part of the total mass flow to still have performance) …

The effect of mass and energy addition in the boundary layer reducing the heat fluxes experimented by the wall can be estimated through semi-empirical results like Rubesin or Kays and Crawford.

CFD can nowadays be used in addition to semi-empirical correlations for the design of such cooling systems, even if some issues exist, especially with commercial codes [61][62].

The effect of coolant nature (due to its specific heat mainly) can also been investigated by CFD, as on the example below.

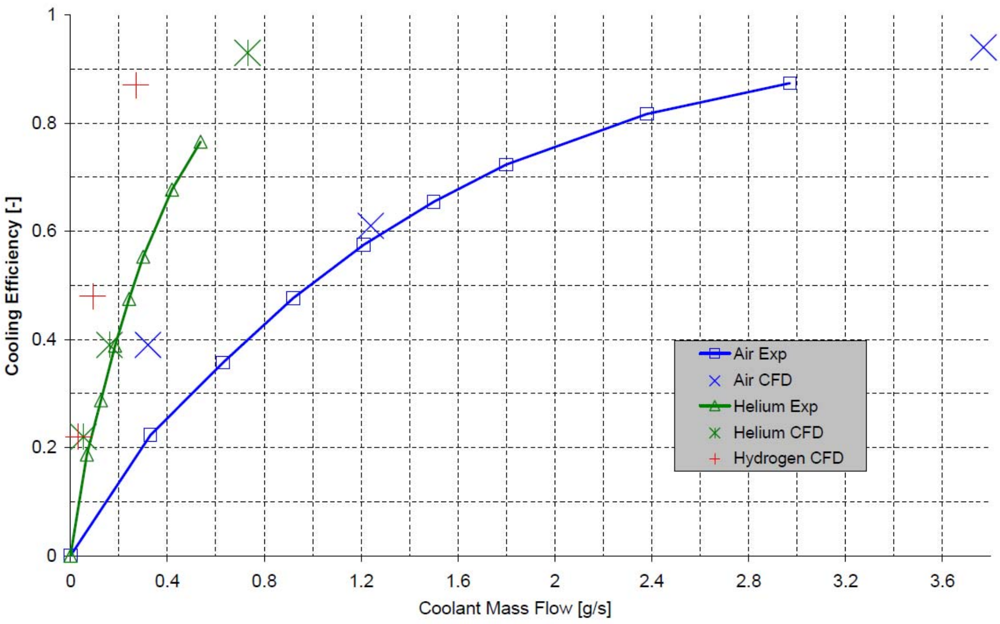


Figure 15: Effusion cooling efficiency in supersonic flow measured at ITLR and computed by ESTEC for different fluids.

Different approaches can be used to compute the heat transfer in the porous medium that constitutes the hot wall of the scramjet. Examples are given below on the supersonic basic ITLR experiment [61][62][60].

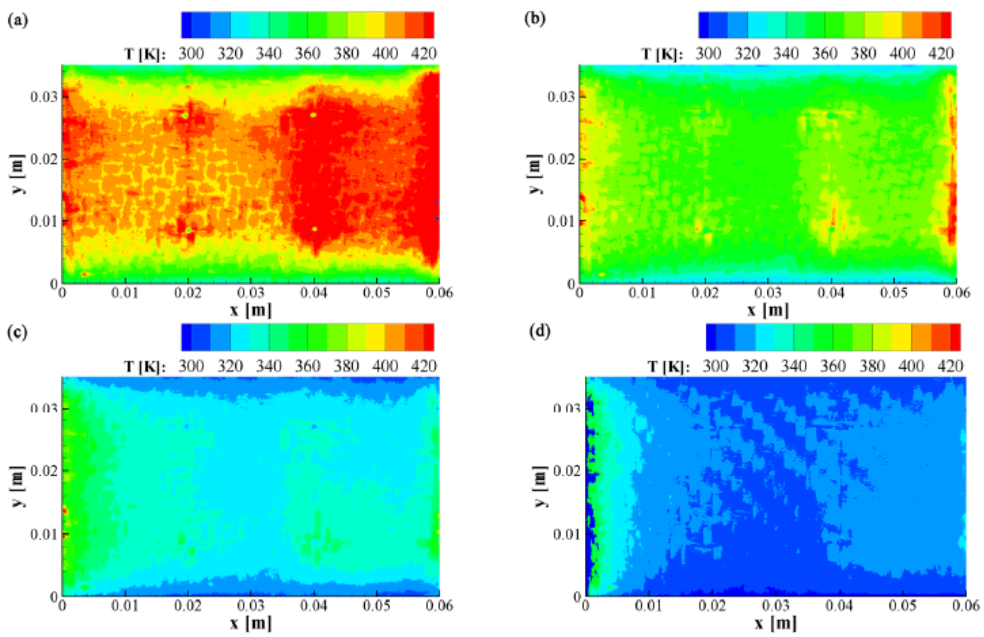


Figure 16: ITLR experimental work on supersonic flow with transpiration wall cooling in ATLLAS 27.

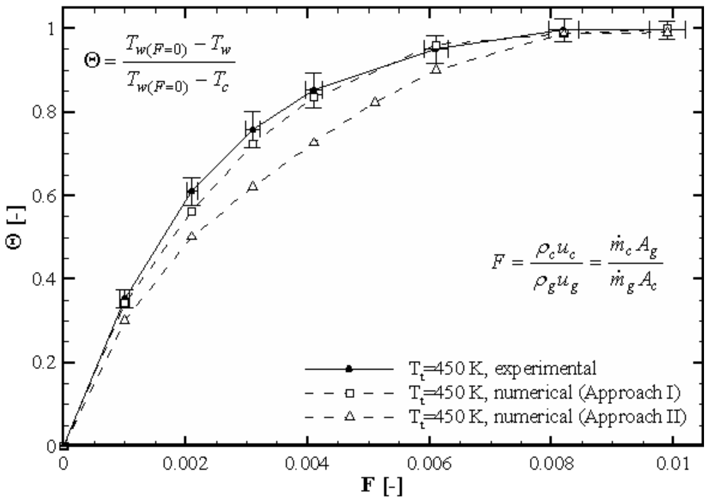


Figure 17: Comparison of two numerical approaches for ITLR transpiration experiment.

Examples of Cooled Structures

Advanced cooled structures have been studied worldwide for application to heat exchangers, high speed vehicles, scramjets and dual-mode ramjets (DMR) (subsonic then supersonic combustion) [14] [26] as well as future liquid rocket engines (LRE) [28][25].

They use high temperature materials, metallic and more and more composite (C/SiC, SiC/SiC, C/C/SiC…).

Different cooling techniques are used. To achieve performance and to limit the risk, the cooled structures are combining these different existing possibilities, leading often to complicate and costly structures.

For example, the channels have to accommodate the change of height of the flow-path (an advantage of pin fin approach is to avoid these confluence problems). Local overheating due to local burning or shock – boundary layer interaction lead to local heat picks, that should be locally solved by local enhancement of the heat transfer or injection cooling.

Metallic Cooled Structures

Often derived from the technology developed for liquid rocket engines, some metallic cooled structures are used, especially if the flight Mach number is “limited”, for example at Mach 6.5.

Two vehicles have flown with scramjet engines cooled by the fuel with such metallic technology: the Kholod axisymmetric hydrogen-fuelled dual-mode ramjet [67] and the hydrocarbon-fuelled X51 demonstrator [51][52].

Advanced Metallic Cooled Thermal Protection

Within the scope of WRR project, more than 30 variants of designs for actively and passively cooled elements heat protective wall panels (“HPE”) had been tested in MAI centre since 1994 [69].

Due to the WRR *Prototype* definition [5] [8], the most important parameter of heat protecting elements HPE (after demonstration of its capacity to withstand the heat flux with the available fuel mass flow) is the weight per unit area. MAI approached a metallic HPE specific weight lower than 12 kg/m2 (assumption of

the system studies, as shown in [8] and [50]). But the technology is quite complex (twin-deck construction, pin fin and internal turbulence enhancement, advanced metallic materials) [14].

Other advanced metallic techniques are described for example in [26][10][11][71].

Ceramic or Metallic/CMC Composite Structures

Many technologies are under investigation, to take benefit of the high temperature capability of the Ceramic Matric Composite materials (1850 K demonstrated during several minutes in oxidative environment within PTAH-SOCAR work) : C/C/SiC or C/SiC.

1) Some of them associate metallic tubes or panels with CMC hot structures in Europe, USA, Japan, China … [1][2][25][33][46][47][54][56][59][61] [71].

2) Others composite cooled structures are based on the linking of different CMC materials.

In this class of architecture, the channels are machined in the CMC panels which are then fixed together.

The joining can be done:

- By screws or by “sock” principle.
- By special gluing like in the Saint Elme demonstration [19] (Figure 41).
- By brazing like in the A3CP technology [56][57][60].

3) One of the problems to be solved is the management of the 2D shapes of most of the scramjet ducts (assembly of panels on corners where the gas pressure load lead to maximum stress). Monobloc composite structure has then an interest compared to the two previous techniques. PTAH-SOCAR technology is one example of this monobloc cooled CMC approach.

Composite PTAH-SOCAR Technology

This In-house R&T effort of MBDA FRANCE and EADS leads to a low cost, highly reliable, effective Fuel-Cooled structure technology, called PTAH-SOCAR. The patented idea has been to develop and preliminary check a concept of C/SiC structure with the following advantages:

- No bonding system (nor brazing, nor gluing…).
- Complete combustion chamber structure in one part (“monobloc”).
- Limitation of connecting problems.
- No problem for realizing corners of a 2D combustor.
- Limitation of possible leakage problems.
- No need of machining internal channels.
- Easier integration of specific systems (injectors, flame-holders..).

As shown on Figure 18 the main ideas for the manufacturing of a whole DMR engine with PTAH-SOCAR are the following:

- Monobloc actively cooled combustion chamber obtained at preform state before its densification process (whatever this one : C/C or C/SiC, CVI or LSI).
- Linking by stitching of complex woven performs.

- Hot and cold skins linked together by stitching with carbon yarn.
- Stitching treads go through the cooling channel.
- Back structure needed to hold the combustion chamber pressure (may be external or integrated at preform state, based on carbon honeycomb, corrugated skin or a system of O3S assembled stiffeners).

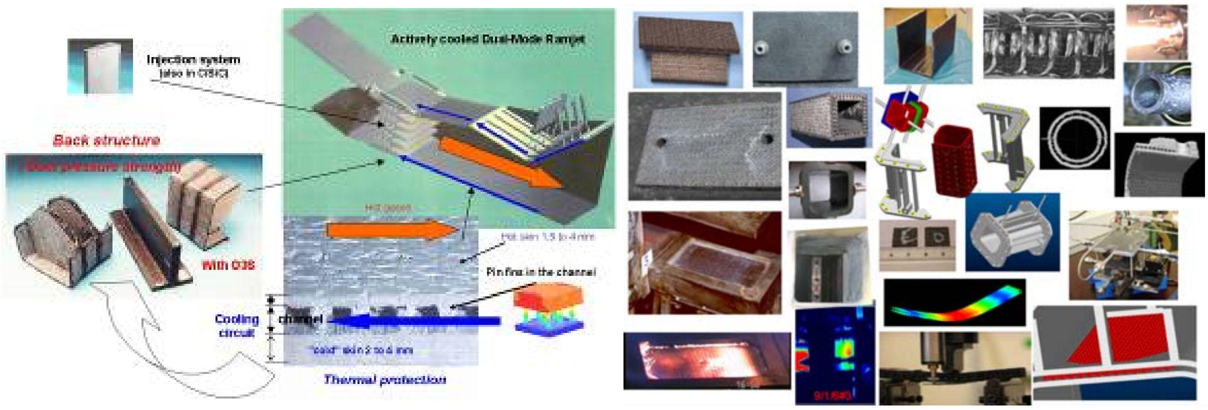


Figure 18: The monobloc CMC technology : PTAH-SOCAR.

The PTAH-SOCAR specific weight for the heat protection system is lower than 10 kg/m2 (density of this CMC material is closed to 2000 kg/m3). With the back structure, the total specific weight is 30% lighter than metallic advanced cooled structures.

The thermal behaviour of different PTAH-SOCAR cooled panels has been checked in scramjet environment during hot test series, with decreasing mass flows of coolant (gaseous nitrogen or liquid kerosene) [15]. Maximum wall temperature was over 1800 K without damage and the cumulative duration of hot tests was 5 minutes. The thermal design and associate models were demonstrated (Figure 18).

The necessary models of the cooled structure and the associated feasibility were checked on the basis on gaseous densification, leading to C/C or C/SiC cooled structures.

The period 1999/2001 was used to check with limited amount of funding and aggressive time schedule the key-points of the PTAH-SOCAR technology.

The following periods have been allowing to secure a cost-effective densification technique for CMC and test more and more pieces in relevant environments (successful cycling testing of scramjet ducts and panels, high pressure tubes, ..).

Demonstrated maximum temperature of hot wall is 1850K, demonstrated internal pressure of coolant is over 190 bar.

Details can be found in [15][16][29][30][31][35][36][40][53].

Engineering of Coupled Phenomena in Cooled Scramjets

Such a system is highly coupled, especially in case of active cooling.

The study of the cooled structure is also a typical illustration of multi-physics coupled phenomena, as the figure below tries to illustrate [32].

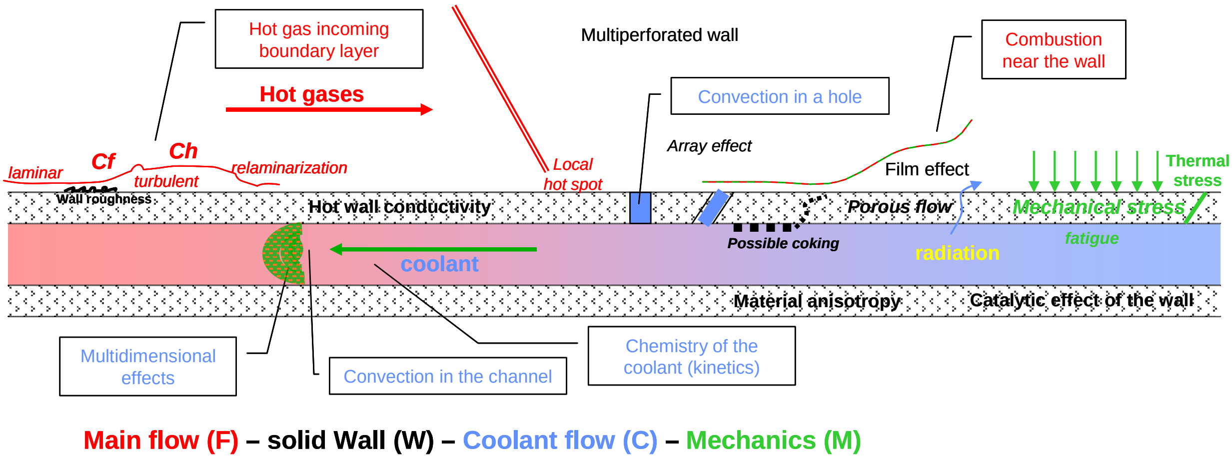


Figure 19: Coupled phenomena in cooled structures.

Information can be found for example in [32] [60][61][62]. The approach associates:

- Engineering codes like NANCY, CASTOR-FEM and semi-empirical data.
- Detailed analysis with 3D/2D codes like CFD-ACE, SAMCEF, FLUENT, CFX, CEDRE, …
- Multiphysics and conjugate problems…
- Coupling of codes or multi-physic solver.
- Step-by-step validation methodology to ensure correct prediction.

Fuel Heat Sink Management

Hydrogen is clearly the best fuel for scramjet, considering the heat release as well as the cooling capability (heat sink). But its density and storage conditions are not the best ones for the vehicle designer.

For Mach 12 flight, the hydrogen mass flow is enough to cool the engine, despite the very high heat loads, and thanks to the high physical heat sink of hydrogen (15 MJ/kg). It is interesting to inject the fuel, as much as it is made possible to still have a good mixing process, in the axial direction, to take benefit of the fuel axial momentum, compensating the heat losses through the cooled walls.

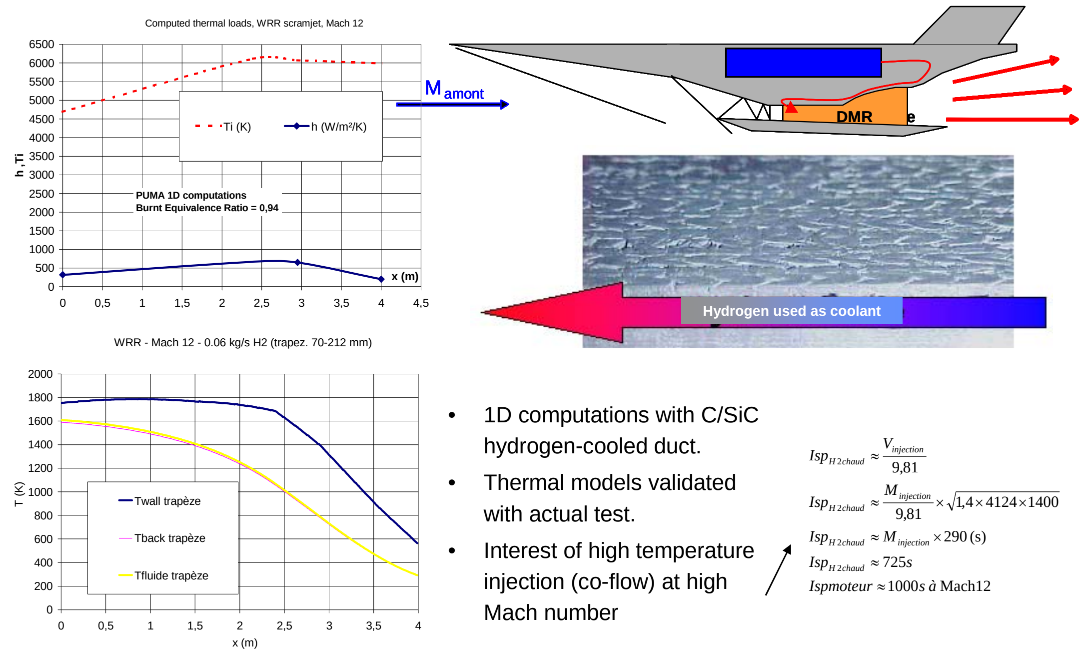


Figure 20: Example of regeneratively cooled DMR (Mach 12).

Up to Mach 8 flight conditions, if a storable hydrocarbon is used, like the American JP7, the Russian T15, more or less regular kerosene or synthesis fuels (China “number 3”, JP8, JP10, …), the physical effect of heat sink (about 1 MJ/kg) is not enough. But a maximum value of 5 MJ/kg can be reached if the pyrolysis is mastered, leading to endothermic decomposition in the cooling circuit but not to excessive coke deposit in the channels or in the injection ports. Need (fuel heat load) and capability (fuel heat sink) is below:

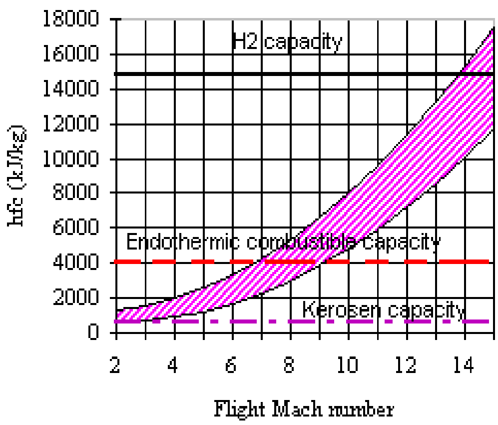


Figure 21: Typical DMR head load with Mach number and heat sink capability of different fuels.

Of course this hydrocarbon capability depends on the nature of the fuel, of the temperature history, of the local speed (residence time), noting that the fuel density can vary by two order of magnitudes during its heating. An example of the residence time and temperature in a scramjet regenerative circuit, and the following figure shows the computed composition at the exit (100% decomposition for this example) can be found in [49].

Then, the basic work on endothermal fuels deals with experiment with different types or reactor as well as residence time and possible solid or liquid catalyst. The table below shows different types of such experiments, examples from literature below were discussed during the course.

Table 2: Experiments on endothermic fuel cooling: overview.

Type of experiment	interest	measurements
Perfectly stirred reactor	Detailed validation of kinetic modeling	Detailed analysis (GC/MS)
Tube with 1 g/s of fuel	Thermal and pyrolysis with actual residence time and simple circuit	Detailed analysis (GC/MS)
Tube with 0,1 g/s of fuel	Compare different process and fuels Define on-line measurement systems	Detailed analysis (GC/MS) FTIR
Autoclave	Compare different process and fuels Thermal or catalytical pyrolysis	Simplified analysis
Small panels or ducts	Thermal behaviour, endurance with different fluids	Thermocuple and pressure, generally no chemical analysis
Real panels or engine components	Real size behaviour	Thermocuple and pressure, generally no chemical analysis

For example, the MPP experiment at ONERA allows studying decomposition and basic supersonic combustion of fuel with a residence time corresponding to the ones expected in actual scramjet cooling systems, showing the influence of the main parameters (heating bloc temperature, fuel pressure and mass flow) on the temperature and composition at the exit of the MPP tube or of the heated transfer line (TL) used to carry the heated fuel (norpar) to the LAERTE basic combustion experiment [43].

Other results, mainly from USA, Russia and recently China can be found, for example in [1][33][42][43][51][59][64][65][66].

Modelling effort is also necessary. For example MBDA uses a detailed pyrolysis kinetic model of dodécane C12H26 developed with CNRS laboratory at NANCY [49].

This kind of model is not designed to adress coke deposition. A first approach to solve this coking prediction problem is to focus on C6H6 (as benzene is known to be a major coke promoter). If we consider that all C6H6 will turn into coke, the overall coke formation will be over-estimated and system studies could be performed on such basis. Heterogeneous reactions (catalyst from the wall, solid carbon formation) can also be documented and added if necessary in the pyrolysis modelling : coke deposit could be a main issue in the case of long cruise flight, reusable engine, but probably not in the case of short flight-time (10-15 min), non-reusable engine.

These basic experiments in the first lines of table 2 allow also comparing different fuels or model-hydrocarbons:

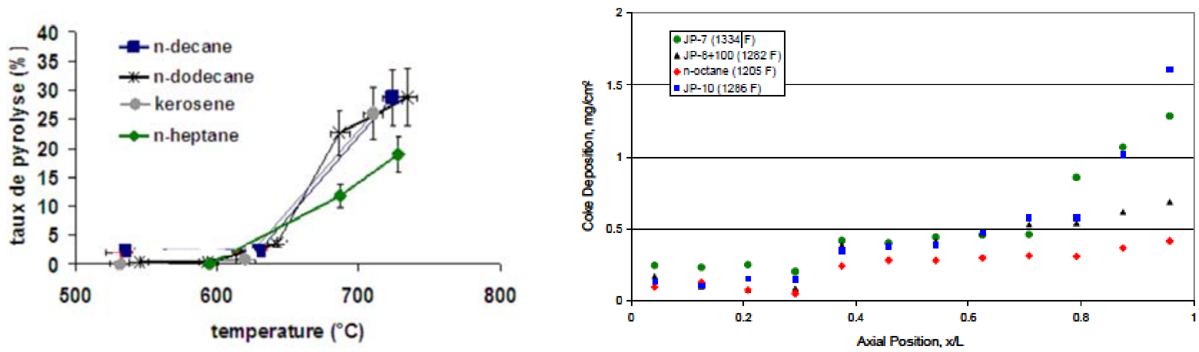


Figure 22: Basic comparison of fuels pyrolysis
(left: decomposition rate in [44], right: coke deposit in [48]).

Some work, like in the COMPARER project, aims at investigating instrumentation techniques (for example IR signal from hot fuel in the cooling channel) to be used to characterize, on-ground and possibly in-flight , the decomposed fuel to be injected [44].

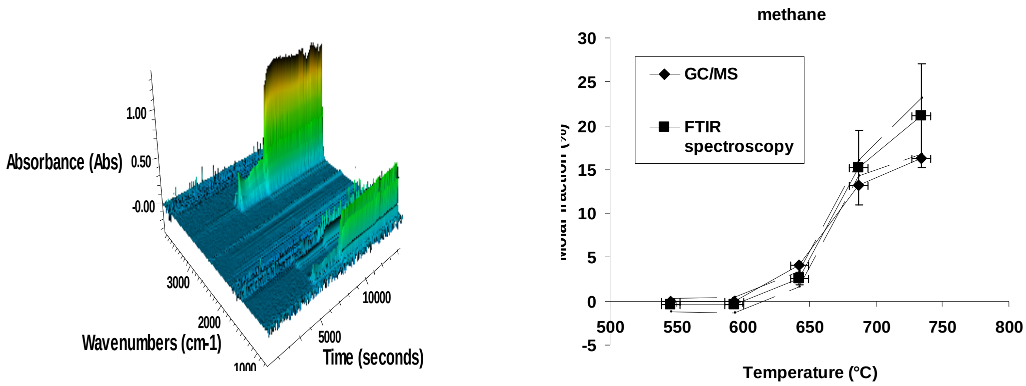


Figure 23: Laboratory demonstration of FTIR optical analysis of decomposed endothermic fuel [44].

Leading Edge and Stagnation Points

Uncooled Leading Edges

For the external leading edge of the vehicle, as well as for the wings or the air intake, uncooled leading edges have been studied and demonstrated. Different materials can be used : high conductivity and high temperature metallic alloys like passivated molybdenum or tungsten, C/SiC CMC composites, Ultra High Temperature Ceramic composites [61]. It should be noticed that in case of high conductivity, the decrease of leading edge radius can lead to lowering its temperature, because of 2D/3D conduction pick-up from the stagnation point to the lateral ones.

This technique can sometimes be used, at “moderate” hypersonic speeds (flight Mach number of 6 for example), for the internally-located injection struts.

Regenerative Cooling of Leading Edges

Different techniques can be used, generally the same as the ones used in blades cooling [23].

For example, the code CHARDON was another result of a many-years cooperation between MBDA and education/research institutes [11]. This engineering code is able to simulate the active cooling of leading edges and had been used in particular to design the cooling system of the St-Elme strut (Figure 24) [18].

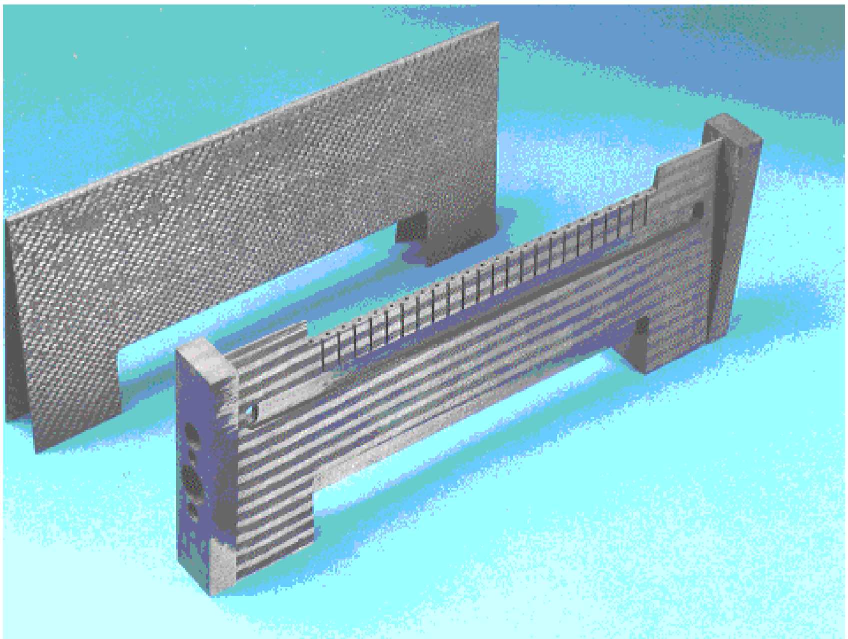


Figure 24: St-Elme carbon/carbon strut components.

The impingement cooling of the leading edge stagnation point can be seen on the body on Figure 24.

FEM thermo-mechanical computations with anisotropy and temperature dependant characteristics (including the Protection Against Oxidation) have demonstrated that the stress is affordable by the material and that the CHARDON engineering code over-estimated the leading edge maximum temperature by 200K [18].

The Saint Elme project allowed to test this protected C/C technology in the Bourges' hypersonic test facility (now called METHYLE) [18][34][33].

Transpiration Cooling of Leading Edges

Within the scope of WRR partnership, tests of metallic leading edge samples (the radius of bluntness is 1.5 mm) have been conducted under strong shock/shock interaction conditions. Their results permit to be confident in the capability of the optimised porous leading edge to work up to Mach 12 conditions.

Figure 25: WRR transpiration-cooled metallic leading edge during test.

The combination of high temperature composite with porous leading has been partially investigated during St-Elme [18] and PTAH-SOCAR studies. The potential interest has been computed, but the corresponding technology cannot yet be considered as demonstrated.

Some activities are in progress especially with DLR technology to investigate more deeply the actual capability of using porous CMC leading edges for injection struts of scramjets [61].

Demonstrated Minimum Radius of Scramjet Injection Strut Leading Edge

The technological work and associated studies from MBDA and their partners allowed giving on Table 3 a summary of the minimum radii for injection struts or pylons of DMR, operating up to Mach 12 with GH2 as coolant.

Table 3: Demonstrated radius for scramjet injection struts leading edges.

	metallic	**composite**
Example of Material	Narloy-Z Inconel	Carbon-Carbon, Protected Against Oxidation
Convective cooling (impingement)	3 mm	1.7 mm
Transpiration/effusion	1.5 mm	Not yet demonstrated

The technological effort can lead to secure these data that can currently be used for the aerodynamic design of scramjet injection systems.

EXERCISE ("BY HAND" CALCULATION)

Generic Scramjet Considered

A generic scramjet has been set to be able to do "by hand" simple estimating of such thermal management.

The considered flight Mach number is Mach 7. This generic scholar scramjet was used with simple calculations to investigate the thermal heat fluxes, the cooling requirements, and compare different solutions (metallic or ceramic composite structures) or possible strategy of cooling of the complete engine (air intake, combustor and nozzle).

The corresponding mass flows are, for conditions close to the ones described on Figure 3, Figure 4, and Figure 5, the following:

- Air mass flow about 7 kg/s.
- Hydrocarbon fuel mass flow: 0.5 kg/s at ER=1 (acceleration).
- Cruise conditions assumed at ER=0.6: fuel mass flow: 0.3 kg/s.

Estimation of the Heat Loads

For the burnt gases, a Cp/cv=1.26 and r=287 J/kg/K can be taken into account.

The simple technique summarized in §2.3 can be used to estimate the heat loads in the air intake (taken at the entry of the combustor), in the combustor (taken at the exit, where local Mach number is 1.4 and static pressure raised up to 1 bar (use Figure 3, Figure 4 and Figure 5).

The computed heat flux densities are:

- 720 W/m²/K *(2148 K – Twall) in the air intake (0.8 MW/m² if Twall=1000K).
- 519 W/m²/K * (3073 K – Twall) in the combustor (1.1 MW/M² with Twall=1000K).

After some additional computation (isentropic expansion from the exit of the combustor up to the nozzle exit chosen area), the same estimating can be done in the nozzle.

Estimation of Different Solutions for Active Cooling

The computed heat loads can then be used, after a security-uncertainty treatment (here we took 30 % more than the Colburn formula, to estimate the cooling strategy of the considered scramjet.

We first consider only the combustor cooled with the available fuel (H/C or H2), with water (ground testing), alone or with air intake and nozzles (in this case we should reserve a part of the coolant flow and use for example only 58% for the combustor). The water mass flow was chosen to avoid local boiling.

This estimating highlights the big heat sink capacity of H2 as well as the possible problem due to the moderate hydrocarbon heat sink.

The following example deals with acceleration phase, at Mach 7, with a chosen repartition of fuel flow for the cooling of the air intake (co-flow heat exchanger), the combustor (counter flow cooling) and part of the nozzle (the other part is assumed to be radiatively cooled). For this case the total hydrocarbon fuel mass flow available is 0.5 kg/s (ER = 1).

The cruise condition can also be estimated. We assume that, due to the lower equivalence ratio, the hot gas temperature is 20 % lower, and that the heat transfer coefficient was not changed enough to be taken into account (mostly dependant on the total gas flow). The total mass flow is here only 0.3 kg/s.

With metallic structure, we are close to the limit of the hydrocarbon fuel (5MJ/kg), then it is interesting to estimate the benefit we could have while using a CMC technology, leading to a wall temperature of 1800K. Thus the heat load of the fuel leads to more practical values, for example 2.4 MJ/kg for the hydrocarbon-cooled combustor.

Table 4: Result for metallic or CMC composite scramjet combustor.

Heat transfer coefficient	675 W/m²/K	
Hot gas adiabatic recovery temperature	2460 K	
Type of structure	*Metallic*	*CMC*
Associated hot wall temperature	*1100 K*	*1800 K*
Heat flux density	917 kW/m²	444 kW/m²
Combustor wetted area	0,95 m²	
Fuel mass flow available to cool the combustor	0.175 kg/s	
Fuel heat load	5 MJ/kg	2.4 MJ/kg

This exercise just wanted to help the engineer or the scientist to manipulate simple formulae and figures in order to have better feeling of the scramjet thermal management issues.

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